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AEROELASTIC RESEARCH AT HIGH SPEEDS

By A. Gerald Rainey

Langley Research Center Langley Field, Va.

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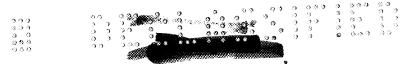
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AEROELASTIC RESEARCH AT HIGH SPEEDS*

By A. Gerald Rainey

SUMMARY

Some of the influence of a few configuration variables on various aeroelastic phenomena at low supersonic and hypersonic speeds has been indicated. It has been shown that a need still exists for a better definition of the limits of applicability of various aerodynamic theories. In addition, there are indications that flutter margins for Dyna-Soartype vehicles will have to be carefully examined at hypersonic speeds as well as in the always troublesome transonic range.

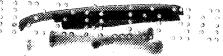
INTRODUCTION

The rigorous requirements imposed on vehicles designed for manned reentry have created aerodynamic and structural conditions which have strained the boundaries of aeroelastic knowledge. New aeroelastic problems have arisen and, unfortunately, not many of the old problems have been abated by these new design requirements. In this paper, some of the current research pertinent to the aerodynamic aspects of these aeroelastic problems is reviewed.

Figure 1 has been prepared to aid in describing some of these problem areas and to delineate the particular aspects discussed in the present paper. The curves drawn in this figure represent typical conditions for a Dyna-Soar-type vehicle in terms of the parameters of most direct interest to the aeroelastician - dynamic pressure and Mach number. Two of the curves represent normal operating conditions - nominal ascent and descent - while the third boundary at higher levels of dynamic pressure represents a design condition for the reentry vehicle which arises from consideration of the requirements for recovering from an off-design trajectory. It is clear that the most severe aeroelastic environment imposed on the reentry vehicle itself is represented by these off-design conditions. The considerations that determine these limit conditions are very interesting. Briefly, at very high velocities these conditions are determined by consideration of the maximum structural







temperatures the vehicle is capable of sustaining, while at lower velocities a relatively arbitrary design limit in dynamic pressure is established. At the intersection of the constant-dynamic-pressure curve and the temperature-limit curve, the combination of maximum dynamic pressure and maximum structural temperatures could create a severe aero-elastic problem area. All along the constant-dynamic-pressure limit, the problems of classical flutter, panel flutter, control effectiveness, and, toward the lower end, transonic flutter must be examined carefully. For the complete booster-vehicle system, the design point for the problems of system stability and atmospheric induced loads occurs in the Mach number range from about 1.0 to 2.0.

The status of the panel flutter problem is discussed in reference 1. A cross section of the literature dealing with the analytical treatment of aeroelastic problems at high speeds is contained in references 2 to 12.

The transonic regime will not be treated at length. This is not to imply that there are no problems here. The critical aeroelastic design conditions may still occur in this speed regime. But in contrast to the other speed regimes, a broad background of experience exists in this area which has provided well-founded procedures for handling these problems (refs. 13 to 15).

This paper consists of two parts. The first part is concerned with some considerations of the structural dynamics problems associated with the boost phase - particularly the aerodynamics involved - and the second part is concerned with the problem of flutter at hypersonic speeds.

SYMBOLS

Ъ	semichord
С	chord
c_n	section normal-force coefficient
E	Young's modulus
EI.	bending stiffness
М	Mach number
\mathbf{q}_{D}	dynamic pressure for divergence



t/c thickness-chord ratio

V velocity

x/l distance from nose as a fraction of body length

 α_{BASE} angle of attack of booster at base

Λ leading-edge sweep angle

μ mass-ratio parameter

 ω_{α} natural frequency in pitch

ω_h natural frequency in translation

AERODYNAMIC CONSIDERATIONS OF STRUCTURAL DYNAMICS

PROBLEMS DURING BOOST

Lifting Surfaces

One of the problems associated with the boosting of large winged payloads is the loss in static stability associated with the elastic deformations of the system. This is not a completely new problem in that several failures of transonic aerodynamic research rockets fired at NASA Wallops Station a number of years ago were traced to this phenomenon (ref. 16). Figure 2 illustrates some recent results obtained in a study of this divergence problem for aerodynamically unstable booster-vehicle systems. The curves in the figure are calculated stability boundaries in terms of the dynamic pressure at which the system becomes statically unstable as a function of the control-system effective stiffness. The results were obtained for a model supported in the manner shown schematically in figure 2. The booster is considered to be rigid and is restrained in pitch by a spring whose stiffness simulates the static aspects of a gimbaled engine control system.

The three curves in the figure represent stability boundaries calculated by means of low-aspect-ratio theory (ref. 3) for three different values of the distributed stiffness. The upper radial line represents the boundary along which a completely rigid vehicle would become unstable. The intermediate curve shows the boundary for a vehicle having a representative stiffness typical of Dyna-Soar designs. The lower calculated boundary may be compared with three measured points obtained at transonic speeds for wind-tunnel models whose stiffness was deliberately lowered to facilitate a study of these aeroelastic effects.





The deviation of the curve for the representative stiffness from the rigid boundary indicates that these static aeroelastic effects can intensify the already severe requirements on a control system for these aerodynamically unstable booster-vehicle systems. Some further information concerning this divergence problem is provided by the results contained in figure 3. These results summarize a part of a systematic study of the divergence characteristics of a series of slender delta wings which were supported as cantilever beams from the trailing edge (ref. 17). The results are plotted in terms of a nondimensional divergence parameter which for a given vehicle would simply be proportional to the dynamic pressure. The cantilever restraint corresponds to an infinitely stiff control system - that is, one which does not permit deviations from the guided path. In general, the relieving effects of Mach number associated with the reduction in bending moments about the trailing edge are predicted by the linear theories and by first-order piston theory. In the low supersonic range where this problem is of most direct concern, the lifting-surface theory agrees fairly well with the experiments. Tests on more highly swept delta wings (see ref. 17) indicate that the lifting-surface theory gives good predictions in the low supersonic Mach number range at sweep angles as high as 80° but begins to deviate seriously from the experimental results at a sweep angle of 85°.

Bodies of Revolution

The preceding results are primarily concerned with the aerodynamics associated with deformed lifting surfaces. Methods for handling the aerodynamics arising from elastic deformations of bodies of revolution are also a question of practical concern in the analysis of structural dynamics problems of boosters. Some results of a study of the pressure distributions on deformed and undeformed bodies of revolution are shown in figure 4. The results are presented in the form of the distribution of section normal-force coefficient for a deformed two-stage booster configuration as indicated at the top of the figure. The circular test points are for the deformed booster with its base at zero angle of attack so that the points represent a direct measure of the section normal-force coefficients produced by this particular deformation. As expected, most of this loading occurs on the conical portions of the body.

One commonly used method of incorporating aerodynamic forces in structural dynamics calculations is to measure the pressure changes with angle of attack on a rigid model and reduce these to local section derivatives which are then applied to angle-of-attack changes regardless of how they are produced - that is, whether by rigid-body motions or by deformations. In order to examine the validity of this procedure,



the square test points were predicted from data obtained from measurements on an undeformed body. Comparison of the circles and squares indicates that this commonly used procedure would have given a fair estimate of the loading due to the deformation of this body in all regions except in the region of the transition section where separation effects predominated.

The easily applied momentum theory (ref. 3) gives only a mediocre estimate of the loading due to the deformation of this body and, unfortunately, more accurate aerodynamic theories are not readily applied to deforming arbitrary bodies.

The preceding sections have been concerned with static aeroelastic effects. Of equal interest are the aerodynamics produced by oscillatory motions. Some information pertinent to this problem is contained in the results of a study of the flutter of cones which is briefly summarized in figure 5. This figure shows some flutter boundaries for a cone supported in such a way that it could pitch and translate on the sting-supported springs shown schematically in the figure.

The stability boundaries are plotted in terms of a nondimensional flutter-velocity index which contains the flutter velocity and density and certain properties describing the dynamics and geometry of the model. For a given vehicle this parameter can be thought of as being proportional to the equivalent airspeed or simply the square root of dynamic pressure. In figure 5, these flutter velocities are plotted as functions of the ratio of the translational frequency to the pitch frequency. These frequencies were varied by adjusting the two springs in the simple support system. The two sets of data points shown in the figure refer to experiments at Mach numbers of 2 and 7. It is interesting to note that both the experiments and the calculations indicate that the minimum flutter velocity is essentially independent of Mach number. This is probably due to the fact that the lift and moment characteristics of cones are relatively insensitive to Mach number changes. The flutter boundary is very sensitive to frequency ratio and illustrates once again the undesirable effects of coupled modes approaching each other.

The calculated curves in figure 5 follow the trends of the rather widely scattered experimental points fairly well. The Newtonian theory (refs. 8 and 9) at hypersonic speeds is directly applicable in aero-elastic analysis of deforming arbitrary bodies, but, unfortunately, the adaptation of a low supersonic cone theory by Von Karman (refs. 2, 8, and 9) to this more general case would be very difficult. Simple momentum theory is more amenable to aeroelastic analysis in this low supersonic speed range, but calculations based on this aerodynamic theory gave very poor agreement in this application.





FLUTTER AT HYPERSONIC SPEEDS

Mach Number Trends

In this section some of the recent information obtained in various studies of the flutter characteristics of lifting surfaces at hypersonic speeds will be considered. One fundamental consideration, of course, is the question as to the probability of a given vehicle encountering flutter at hypersonic speeds rather than at some lower Mach number. This question can be discussed with the aid of figure 6 which shows the variation of the flutter-velocity index with Mach number for a few structural and aerodynamic configurations. It may be recalled that the flutter-velocity index for a given vehicle may be thought of as being proportional to the equivalent airspeed. In addition, it should be pointed out that the use of the structural characteristics, such as mass and natural frequency, in this normalized velocity parameter tends to eliminate structural effects from the data so that the aerodynamic effects of configuration changes are more readily apparent.

All the curves exhibit a minimum flutter speed in the transonic range; however, for two of the configurations there is a trend toward low flutter speeds at the higher Mach numbers. The scatter in flutter points for one of the configurations shown is attributed to the ill-behaved structure used which consisted of a thin flat plate of low aspect ratio clamped along its root to the inside of a cylindrical test section. Similar trends toward low flutter speeds have been obtained by the Cornell Aeronautical Laboratory under an Air Force sponsored project.

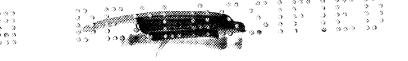
Trend studies of this nature merely serve as a guide in determining where emphasis should be placed in studying the flutter margins of a vehicle. For a Dyna-Soar-type vehicle, which is required to be flutter free at constant equivalent airspeed over a wide range of Mach number, the flutter characteristics must be examined carefully at hypersonic speeds as well as at transonic speeds.

Aerodynamic Effects of Some Geometric Variables

The uncertainties regarding flutter behavior at hypersonic speeds have provided the motive for a variety of investigations. Figures 7 and 8 summarize different parts of a program designed to provide a better understanding of these hypersonic aeroelastic problems.

<u>Leading-edge-bluntness effects.</u> Figure 7 illustrates some results obtained in a study of the effects of leading-edge bluntness at a Mach number of 7. The flutter boundaries are presented in the form of the





flutter-velocity index as a function of leading-edge radius for a series of systematically blunted models supported in such a manner as to permit a rigid-body pitch-flapping type of flutter mode. The models were square in planform and the thickness-chord ratio increased with bluntness so that the included angle between the flat sides remained constant at 10°.

The experimental results shown in figure 7 indicate that the effects of blunting the leading edge are slightly beneficial inasmuch as the flutter speed is increased. For the bluntest model, static divergence rather than flutter was encountered. This fact indicates that the effect of the blunting was probably due to a forward shifting of the center of pressure. The piston-theory result is shown for comparison with the experimental results of figure 7 primarily to illustrate one of its limitations, namely, that even the smallest amount of this type of blunting produces serious violations of its assumptions. Attempts at incorporating the more realistic Newtonian theory show some improvement; however, the lack of agreement in trend for the various theories is somewhat disconcerting. The curve labeled "piston-Newtonian theory" was obtained by applying Newtonian theory to the cylindrical leading edge and piston theory over the remainder of the airfoil in the manner described in reference 2. Perhaps an adaptation of blast-wave theory to this problem might provide an improvement.

Sweepback effects.- Another geometrical consideration of practical concern is the effect of sweepback. A study of the hypersonic flutter characteristics of a series of slender delta wings is summarized in figure 8 (ref. 18). These models, varying in leading-edge sweep angle from 60° to 80°, were supported in such a manner that the flutter mode contained rigid-body pitch and flapping motions.

The flutter boundaries are plotted in terms of the fluttervelocity index as a function of leading-edge sweep angle. In this case, the flutter velocities have been adjusted analytically for a systematic variation of frequency ratio which tended to obscure the aerodynamic effects of sweepback. In addition to variations in the sweepback angle, the models tested also varied systematically in the ratio of flapping frequency to pitch frequency. Since this variation in frequency ratio tended to obscure the aerodynamic effects of sweepback, the values of the flutter-velocity index shown in figure 8 have been adjusted analytically so that they would apply to a case of constant frequency ratio. This was accomplished by multiplying the velocity-index values for each model by the ratio of the value calculated for that model at the frequency ratio of the 700 wing to the value calculated for that model at its own frequency ratio. The uncoupled frequency ratio for the 70° wing was 0.592 and varied over the range from 0.488 for the 60° wing to 0.690 for the 80° wing.





Examination of these adjusted experimental points in figure 8 indicates that the aerodynamic effects of sweepback are very small for this case. The piston-theory results show surprisingly good agreement considering its two-dimensional character. The linear theory shows a trend toward increasing flutter velocities with increasing sweep - a trend which is contradictory to the experimental results.

CONCLUDING REMARKS

Some of the influence of a few configuration variables on various aeroelastic phenomena at low supersonic and hypersonic speeds has been indicated. It has been shown that a need still exists for a better definition of the limits of applicability of various aerodynamic theories. In addition, there are indications that flutter margins for Dyna-Soar-type vehicles will have to be carefully examined at hypersonic speeds as well as in the always troublesome transonic range.

Langley Research Center,
National Aeronautics and Space Administration,
Langley Field, Va., April 12, 1960.



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BOUNDARIES FOR DISCUSSION OF AEROELASTIC PROBLEM AREAS

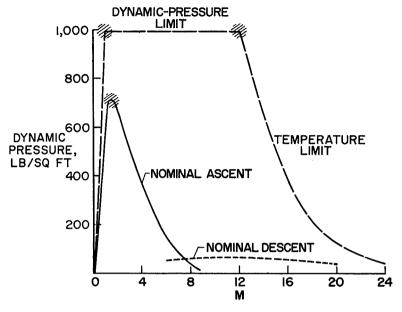


Figure 1

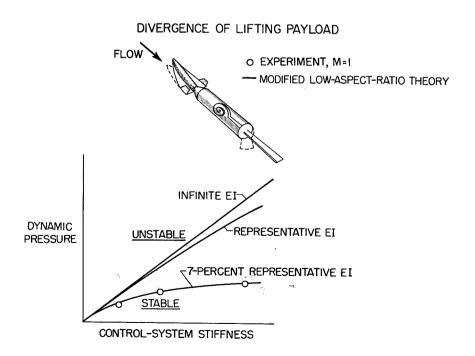


Figure 2



Figure 3

2

DIVERGENCE OF SLENDER DELTA WINGS Λ =70°

O

EXPERIMENT

UNSTABLE

STABLE

6

8

4

M

LINEAR STRIP LIFTING-SURFACE PISTON LOW-ASPECT-RATIO

THEORY

FLOW

 q_D $\frac{\overline{E\left(\frac{1}{C}\right)^3}}{E\left(\frac{1}{C}\right)^3}$

600₁

400

200

0

LOADING DUE TO DEFORMATION M = 1.64

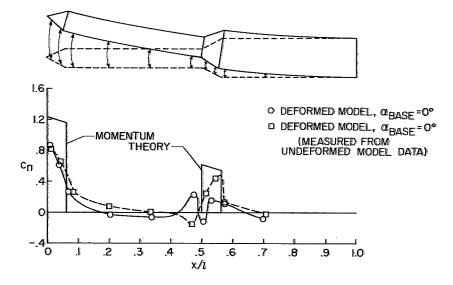


Figure 4



FLUTTER OF CONES

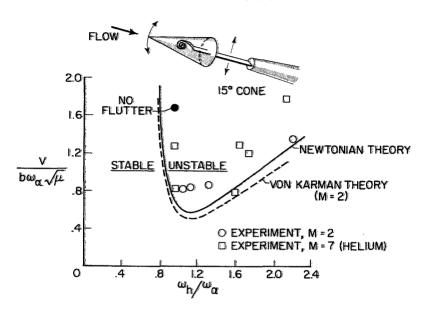


Figure 5

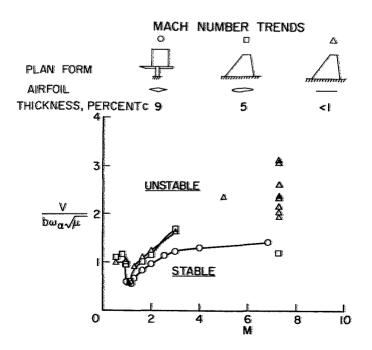
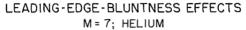


Figure 6





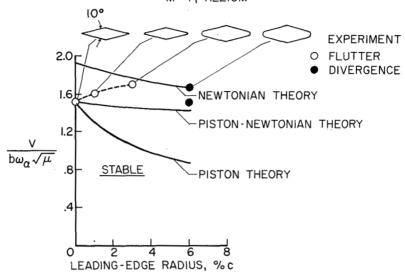


Figure 7

FLUTTER OF SLENDER DELTA WINGS

M=7; HELIUM; 5% THICK, DOUBLE-WEDGE AIRFOIL

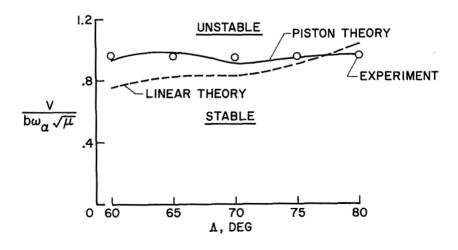


Figure 8



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